CHAPTER 3. DESIGN AND CONSTRUCTION
Section 1. GENERAL

132. SECTION 23.629 (as amended by amendment 23-31) FLUTTER. This subject is covered in AC 23.629-1A.

133.-137. RESERVED.

Section 2. CONTROL SYSTEMS

138. SECTION 23.671 GENERAL. (RESERVED).

139. SECTION 23.677 (as amended by amendment 23-34) TRIM SYSTEMS.

a. Qualitative Evaluation. Trim should be qualitatively evaluated during all phases of the flight test program. Cockpit control trim devices should be evaluated for smoothness, sense of motion, and ease of operation, accessibility, and visibility of the trim tab indicators (both day and night). Ease in establishing and maintaining a trim condition should be evaluated.

b. Electric Trim Background. Electrically-actuated, manually-controlled trim systems have been certificated in several ways, depending on systems design. The simpler systems are tested for failure in flight. More sophisticated systems, which generally incorporate a dual-wire, split-actuating switches, may require a dual failure to produce a runaway. Analysis of these systems discloses that one switch could fail closed and remain undetected until a failure occurred in the other switch or circuit to produce a runaway. This is still considered acceptable if the applicant provided a preflight test procedure that will detect such a dormant failure. Service experience dictates that evaluation of fail-safe trim systems by analysis alone is not acceptable and flight testing is required.

c. Explanation.

(1) Fault Analysis. A fault analysis should be evaluated for each trim system.

(2) Single Failure and Backup System. For a system in which the fault analysis indicates a single failure will cause runaway, flight tests should be conducted. For a system with backup features, or a redundant system where multiple failures would be required for runaway, the certification team should determine the extent of the flight tests necessary after consideration of the fault analysis and determination of the probability and effect of runaway. In all cases, flight test evaluations should be conducted to determine functional system/airplane compatibility in accordance with § 23.1301.

(3) Failure. For the purpose of a fault analysis, a failure is the first fault obviously detectable by the pilot and should follow probable combinations of undetectable failures assumed as latent failures existing at the occurrence of the detectable failure. When an initial failure also causes other failures, the initial failure and the subsequent other failures are considered to constitute a single failure for purposes of fault analysis; that is, only independent failures may be introduced into the fault analysis to show multiple failure integrity.

(4) Failure Warning. The first indication a pilot has of a trim runaway is a deviation from the intended flight path, abnormal control movements, or a warning from a reliable failure warning system. An aural or flashing visual warning signal (in clear view of the pilot), actuated by a trim-in-motion system, is considered to give the pilot clear warning. Consequently, pilot recognition time is considered negligible with a trim-in-motion system. The following time delays after pilot recognition are considered appropriate:

(i) Takeoff, approach, landing - 1 second.

(ii) Climb, cruise, descent - 3 seconds.

(5) Second Set of Controls. If a set of controls and instruments are provided for a second crew member, multi-function systems disconnect or quick-disconnect/interrupt switches, as appropriate, should be provided for both crew members regardless of minimum crew.

d. Definitions.

(1) Disconnect Switch. A switch which is located within immediate reach and readily accessible to the pilot, which has the primary purpose of stopping all movement of the electric trim system. A circuit breaker is not considered to be a disconnect switch.

(2) Quick-Disconnect/Interrupt Switch. A switch or device that momentarily interrupts all movement of the electric trim system, which is located on the control wheel on the side opposite the throttles, or on the stick control, that can be operated without moving the hand from its normal position on the control. The primary purpose of the switch is to stop all movement of the electric trim system.

e. Procedures.

(1) Quick-Disconnect or Interrupt Switch. With a quick-disconnect or interrupt switch, disconnect may be initiated after the delay times given in paragraph 139c(4).

(2) Disconnect Switch. With a disconnect switch, the time delays given in paragraph 139c(4) should be applied prior to corrective action by use of primary controls. In addition to these time delays, an appropriate reaction time to disconnect the systems should be added. When there are other switches in the immediate area of the quick-disconnect, a time increment should be added to account for identifying the switch.

(3) Loads. The loads experienced as a result of the malfunction should normally not exceed an envelope of 0 to +2g. The positive limit may be increased if analysis has shown that neither the malfunction nor subsequent corrective action would result in a load beyond limit load. In this case, careful consideration should be given to the delay time applied, since it may be more difficult for the pilot to reach the disconnect switch.

(4) High Speed Malfunctions. When high speed malfunctions are introduced at V_{NE} or V_{MO}/M_{MO}, whichever is appropriate, the speed excursion, using the primary controls and any speed reduction controls/devices, should not exceed the demonstrated upset speed established under § 23.253 for airplanes with a V_{MO}/M_{MO} speed limitation and a speed midway between V_{NE} and V_D for those airplanes certified with a V_{NE} limitation.

(5) Speed Limitations. The use of a reduction of $V_{NE}/V_{MO}/M_{MO}$ in complying with paragraph e(4) of this section is not considered acceptable, unless these new speeds represent limitations for the overall operation of the airplane.

(6) Forces. The forces encountered in the tests should conform to the requirements of § 23.143 for temporary and prolonged application. Also, see paragraph 45 of this AC.

140. SECTION 23.679 (original issue) CONTROL SYSTEM LOCKS. This subject is covered in AC 23.679-1.

141. SECTION 23.697 WING FLAP CONTROLS. (RESERVED).

142. SECTION 23.699 WING FLAP POSITION INDICATOR. (RESERVED).

143. SECTION 23.701 FLAP INTERCONNECTION. (RESERVED).

144.-153. RESERVED.

Section 3. LANDING GEAR

154. SECTION 23.729 (as amended by amendment 23-26) LANDING GEAR EXTENSION AND RETRACTION SYSTEM. This subject is covered in AC 23.729-1.

155. SECTION 23.735 BRAKES. (RESERVED).

156.-160. RESERVED.

Section 4. PERSONNEL AND CARGO ACCOMMODATIONS

161. SECTION 23.771 PILOT COMPARTMENT. (RESERVED).

162. SECTION 23.773 PILOT COMPARTMENT VIEW. (RESERVED).

163. SECTION 23.777 COCKPIT CONTROLS. (RESERVED).

164. SECTION 23.803 (as added by amendment 23-34) EMERGENCY EVACUATION. This subject is covered in AC 20-118A.

165. SECTION 23.807 (as amended by amendment 23-34) EMERGENCY EXITS. AC's 23.807-2 and 23.807-3 address this subject.

166. SECTION 23.831 VENTILATION. (RESERVED).

167.-175. RESERVED.

Section 5. PRESSURIZATION

176. SECTION 23.841 (as amended by amendment 23-17) PRESSURIZED CABINS. AC 23.841-1 addresses this subject.

177. SECTION 23.843 PRESSURIZATION TESTS. (RESERVED).

178.-188. RESERVED.

CHAPTER 4. POWERPLANT
Section 1. GENERAL

189. SECTION 23.901 INSTALLATION. (RESERVED).

190. SECTION 23.903 (as amended by amendment 23-34) ENGINES.

a. Explanation:

(1) Automatic Propeller Feathering Systems. All parts of the feathering device which are integral with the propeller or attached to it in a manner that may affect propeller airworthiness should be considered. The determination of airworthiness should be made on the following basis:

(i) The automatic propeller feathering system should not adversely affect normal propeller operation and should function properly under all temperatures, altitudes, airspeeds, vibrations, accelerations, and other conditions to be expected in normal ground and flight operation.

(ii) The automatic device should be demonstrated to be free from malfunctioning which may cause feathering under any conditions other than those under which it is intended to operate. For example, it should not cause feathering during:

(A) Momentary loss of power.

(B) Approaches with reduced throttle settings.

(iii) The automatic propeller feathering system should be capable of operating in its intended manner whenever the throttle control is in the normal position to provide takeoff power. No special operations at the time of engine failure should be necessary on the part of the crew in order to make the automatic feathering system operative.

(iv) The automatic propeller feathering installation should be such that not more than one engine will be feathered automatically even if more than one engine fails simultaneously.

(v) The automatic propeller feathering installation should be such that normal operation may be regained after the propeller has begun to feather automatically.

(vi) The automatic propeller feathering installation should incorporate a switch or equivalent means to make the system inoperative. (Also see §§ 23.67 and 23.1501.)

(vii) If performance credit is given for the automatic propeller feathering system, there should be a means provided to satisfactorily preflight check the system.

(viii) Most turbopropeller airplanes are equipped with some type of engine ignition system intended for use during flight in heavy precipitation conditions and for takeoff/landing on wet or slush-covered runways. The engine ignition system may be either automatic or continuous. The purpose of this system is to prevent or minimize the possibility of an engine flameout due to water ingestion. Compatibility with auto-feather systems should be ensured.

(2) Negative Torque Sensing Systems. (RESERVED).

b. Procedures.

(1) Automatic and Manual Propeller Feathering System Operational Tests.

(i) Tests should be conducted to determine the time required for the propeller to change from windmilling (with the propeller controls set for takeoff) to the feathered position at the takeoff speed determined in § 23.51.

(ii) The propeller feathering system should be tested to demonstrate nonrotation at one engine inoperative climb airspeed. The configuration should be:

(A) Critical engine inoperative.

(B) Wing flaps retracted.

(C) Landing gear retracted.

(D) Cowl flaps closed.

(iii) The propeller should be tested in the actual configuration for an emergency descent. A sufficient speed range should be covered to assure that any propeller rotation is not hazardous. In addition, the propeller should not inadvertently unfeather during these tests.

(iv) In order to demonstrate that the feathering system operates satisfactorily, propeller feather should be demonstrated throughout both the airspeed and the altitude envelope since engine failure may occur at any time. Propeller unfeathering need only be demonstrated up to the maximum one-engine-inoperative service ceiling or maximum airstart altitude, whichever is higher. Satisfactory propeller unfeathering should also be demonstrated after a 30-minute cold soak.

(2) Continued Rotation of Turbine Engines.

(i) Means should be provided to completely stop the rotation of turbine engines if continued rotation would cause a hazard to the airplane. Devices such as feathering propellers, brakes, doors, or other means may be used to stop turbine engine rotation.

(ii) If engine induction air duct doors or other types of brakes are provided to control engine rotation, no single fault or failure of the system controlling engine rotation should cause the inadvertent travel of the doors toward the closed position or the inadvertent energizing of braking means, unless compensating features are provided to ensure that engine failure or a critical operating condition will not occur. Such provisions should be of a high order of reliability, and the probability should be remote that doors or brakes will not function normally on demand.

(3) Engine Operation with Automatic Propeller Control System Installed.

(i) When an automatic control system for simultaneous r.p.m. control of all propellers is installed, it should be shown that no single failure or malfunction in this system or in an engine controlling this system will:

(A) Cause the allowable engine overspeed for this condition to be exceeded at any time.

(B) Cause a loss of thrust which will cause the airplane to fail to meet the requirements of §§ 23.51 through 23.77 if such system is certificated for use during takeoff and climb. This should be shown for all weights and altitudes for which certification is desired. A period of 5 seconds should be allowed from the time the malfunction occurs to the initial motion of the cockpit control for corrective action taken by the crew.

(ii) Compliance with this policy may be shown by analysis, flight demonstration, or a combination thereof.

(4) Intermixing of Engines.

(i) Explanation. Engines of different ratings and/or cowls may be intermixed on airplanes provided the proper performance associated with the engine combination is used. In general, for four-, three- or two-engine airplanes, the performance combination is as follows:

(A) When one lower thrust engine is installed, the normal AFM performance level is reduced by an increment appropriate to the decrease in thrust resulting from the intermix.

(B) When more than one lower thrust engine is installed, the performance should be based on the thrust of the lower/lowest rated engine.

(C) The V_{MC} should be based on the highest thrust engine(s).

(ii) Procedures.

(A) The operating procedures should be provided for all engines installed, that is, relight altitude/airspeed, flight idle lights, etc. Differences in operating methods should be limited to the equivalent of having a maximum of two different engines on the airplane.

(B) Air conditioning packs and bleed configurations for takeoff should be such that no more than two thrust parameters are to be monitored.

(C) Only one odd engine or odd operation of the same engines is permitted. A placard should be installed notifying the pilot of the odd case. However, two engines of each kind may be intermixed, but all engine limits and markings should be appropriate to the engine installed.

(D) The number of thrust gauges operative for takeoff should be considered for each individual case.

(E) Temperature and r.p.m. limits for the engine installed or the rating at which it will be operated, should be properly presented to the pilot in accordance with §§ 23.1541 through 23.1543.

c. Restart Envelope. For turbine engine-powered airplanes, the applicant should propose an airstart envelope wherein satisfactory inflight engine restarts may be accomplished. Airstarts should be accomplished satisfactorily at critical combinations of airspeed and altitude. During these tests, normally time history data showing airspeed, altitude, r.p.m., exhaust temperature, etc., are obtained for inclusion in the Type Inspection Report. The airstart envelope should be included in the limitations sections of the AFM. The procedures used to restart the engine(s) should be contained in the emergency or abnormal procedures section of the AFM.

191. SECTION 23.905 (as amended by amendment 23-29) PROPELLERS. Included in § 23.903 material. See paragraph 190 of this AC.

192. SECTION 23.909 (as amended by amendment 23-7) TURBOSUPERCHARGERS. AC 23.909-1 addresses this subject.

193. SECTION 23.929 (as amended by amendment 23-14) ENGINE INSTALLATION ICE PROTECTION.

a. Explanation. This regulation requires that propellers and other components of the complete engine installation such as oil cooling inlets, generator cooling inlets, etc., function satisfactorily and operate properly without appreciable loss of power when the applicant requests approval for flight in icing conditions. See § 23.1093 for induction system ice protection requirements.

b. Procedures. Each propeller and other components of the complete installation that is to be approved for flight in icing conditions should be evaluated under the icing conditions specified in Part 25, appendix C. If the propellers are equipped with fluid-type deicers, the flow test should be conducted starting with a full tank of fluid and operated at maximum flow for a time period found operationally suitable. The operation should be checked at all engine speeds and powers.

194. SECTION 23.933 (as amended by amendment 23-34) REVERSING SYSTEMS.

a. Explanation. Self-explanatory.

b. Procedures. Reverse thrust propeller installations may be approved provided the following is acceptable:

(1) Exceptional pilot skill should not be required in taxiing or any condition in which reverse thrust is to be used.

(2) Necessary operating procedures, operating limitations, and placards are established.

(3) The airplane control characteristics are satisfactory with regard to control forces encountered, and buffeting should not cause structural damage.

(4) The directional control is adequate using normal piloting skill.

(5) A determination is made that no dangerous condition is encountered in the event of sudden failure of one engine in any likely operating condition.

(6) The operating procedures and airplane configuration are such as to provide reasonable safeguards against serious structural damage to parts of the airplane due to the reverse airflow.

(7) It is determined that the pilot's vision is not dangerously obscured under normal operating conditions on dusty or wet runways and where light snow is on the runway.

(8) It is determined that the pilot's vision is not dangerously obscured by spray due to reverse airflow under normal water operating conditions with seaplanes.

(9) The procedure and mechanisms for reversing should provide a reverse idle setting such that without requiring exceptional piloting skill at least the following conditions are met:

(i) Sufficient power is maintained to keep the engine running at an adequate speed to prevent engine stalling during and after the propeller reversing operation.

(ii) The propeller does not overspeed during and after the propeller reversing operation.

(10) The engine cooling characteristics should be satisfactory in any likely operating condition.

(11) If using ground idle for disking drag credit on landing distance, the ground idle position of the power levers should be identified with a gate or a detent with satisfactory tactile feel (reference paragraph 27a(7) of this AC).

195. SECTION 23.939 (as amended by amendment 23-18) POWERPLANT OPERATING CHARACTERISTICS.

a. Explanation. Self-explanatory.

b. Procedures.

(1) Stall, Surge, Flameout Tests. For turbine engines, tests should be conducted to determine that stall, surge, and flameout will not occur, to a hazardous degree, on any engine during acceleration and deceleration throughout the normal flight envelope of the airplane. This would include tests throughout the approved altitude range and throughout the airspeed range from V_S to V_{MO}/M_{MO} using sideslip angles appropriate to the individual airplane. For normal category multiengine airplanes, an appropriate sideslip angle is generally considered to be approximately one ball width on a standard slip-skid indicator. The low airspeed tests should be accomplished at light weight and with gear and flaps extended to further reduce the stall speed. Tests need not be accomplished with gear and flaps extended at airspeeds above which extension is prohibited in the AFM. At the conditions mentioned above, the effects of engine bleed air off and on and engine ice protection systems off and on should be investigated.

(2) Throttle Techniques. With the engine stabilized at maximum continuous power, rapidly retard the throttle to the flight idle position. Before the engine reaches idle power or r.p.m., rapidly advance the throttle to maximum continuous power. Repeat this process except begin with the engine stabilized at flight idle power. Rapid throttle movement is generally defined as one which results in the throttle moving from maximum continuous power to flight idle, or vice versa, in not more than 0.5 seconds.

196. SECTION 23.943 (as amended by amendment 23-18) NEGATIVE ACCELERATION.

a. Explanation. Tests should be conducted to show that no hazardous malfunction occurs under negative accelerations within the flight envelope. A hazardous malfunction in this case usually is considered to be one which causes a loss or sustained malfunction of the engine, or improper operation of the engine accessories or systems.

b. Procedures.

(1) Tests. Critical points of negative acceleration may be determined through tests. Consideration should be given to the possibility of critical levels of fuel and oil.

(2) Flight Envelope. With engines operating at maximum continuous power, the airplane is flown at a critical negative acceleration within the prescribed flight envelope. Normally a duration of the negative acceleration of -0.2g for 5 seconds, -0.3g for 4 seconds, -0.4g for 3 seconds, and -0.5g for 2 seconds should reveal any existing hazardous malfunctioning of the engine. Alternately, -0.5g for 5 seconds may be used. In addition, it may be necessary to consider other points within the flight envelope at other levels of fuel with shorter applications of accelerations. In all cases, the accelerations are measured as near as practicable to the c.g. of the airplane.

197.-206. RESERVED.

Section 2. FUEL SYSTEM

207. SECTION 23.959 (as amended by amendment 23-18) UNUSABLE FUEL SUPPLY. This subject is covered in AC 23.959-1.

208. SECTION 23.961 (original issue) FUEL SYSTEM HOT WEATHER OPERATION. This subject is covered in AC 23.961-1.

209.-220. RESERVED.

Section 3. FUEL SYSTEM COMPONENTS

221. SECTION 23.1001 (as added by amendment 23-7) FUEL JETTISONING SYSTEM.

a. Explanation. The basic purpose of these tests is to determine that the required amount of fuel may be safely jettisoned under reasonably anticipated operating conditions within the prescribed time limit without danger from fire, explosion, or adverse effects on the flying qualities. The applicant should have made sufficient jettisoning tests to prove the safety of the jettisoning system.

b. Procedures.

(1) Fire Hazard.

(i) Fuel in liquid or vapor form should not impinge upon any external surface of the airplane during or after jettisoning. Colored fuel, or surfaces so treated that liquid or vaporous fuel changes the appearance of the airplane surface, may be used for detection purposes. Other equivalent methods for detection may be acceptable.

(ii) Fuel in liquid or vapor form should not enter any portion of the airplane during or after jettisoning. The fuel may be detected by its scent, combustible mixture detector, or by visual inspection. In pressurized airplanes, the presence of liquid or vaporous fuel should be checked with the airplane unpressurized.

(iii) There should be no evidence of fuel valve leakage after it is closed.

(iv) If there is any evidence that wing flap (slats/slots) positions other than that used for the test may adversely affect the flow pattern, the airplane should be placarded "Fuel should not be jettisoned except when flaps (slats/slots) are set at ___ degrees."

(v) The applicant should select, for demonstration, the tanks or tank combinations which are critical for demonstrating the flow rate during jettisoning.

(vi) Fuel jettisoning flow pattern should be demonstrated from all normally used tank or tank combinations on both sides of the airplane whether or not both sides are symmetrical.

(vii) Fuel jettisoning rate may be demonstrated from only one side of symmetrical tank or tank combinations which are critical for flow rate.

(viii) Fuel jettisoning rate and flow pattern should be demonstrated when jettisoning from full tanks using fuel.

(2) Control.

(i) Changes in the airplane control forces during the fuel jettisoning tests should be noted.

(ii) The capability to shut off the fuel jettisoning system should be demonstrated in flight.

(3) Residual Fuel. The residual fuel should be measured by draining the tanks from which fuel has been jettisoned in flight, measuring the total drained fuel, and subtracting from the total the unusable fuel quantity for each tank to determine if there is sufficient reserve fuel after jettisoning to meet the requirements of this section. This may be a ground test.

222.-237. RESERVED.

Section 4. OIL SYSTEM

238. SECTION 23.1027 (as amended by amendment 23-14) PROPELLER FEATHERING SYSTEM. Included in § 23.903 material. See paragraph 190 of this AC.

239.-244. RESERVED.

Section 5. COOLING

245. SECTION 23.1041 (as amended by amendment 23-7) GENERAL. See paragraphs 246, 247, and 248 of this AC.

246. SECTION 23.1043 (as amended by amendment 23-21) COOLING TESTS.

a. Explanation. Paragraphs 247 and 248 of this AC provide details on reciprocating engine and turbine engine cooling tests. Additional procedures for certification of winterization equipment are given below.

b. Winterization Equipment Procedures. The following procedures should be applied when certificating winterization equipment:

(1) Other Than a 100-Degree Farenheit Day. Cooling test results for winterization installations may be corrected to any temperature desired by the applicant rather than the conventional 100°F hot-day. For example, an applicant may choose to demonstrate cooling to comply with requirements for a 50° or 60° day with winterization equipment installed. This temperature becomes a limitation to be shown in the AFM. In such a case, the sea level temperature for correction purposes should be considered to be the value elected by the applicant with a rate of temperature drop of 3.6°F per thousand feet above sea level.

(2) Tests. Cooling tests and temperature correction methods should be the same as for conventional cooling tests.

(3) Limit Temperature. The AFM should clearly indicate that winterization equipment should be removed whenever the temperature reaches the limit for which adequate cooling has been demonstrated. The cockpit should be placarded accordingly.

(4) Equipment Marking. If practical, winterization equipment, such as baffles for oil radiators or for engine cooling air openings, should be marked clearly to indicate the limiting temperature at which this equipment should be removed.

(5) Installation Instructions. Since winterization equipment is often supplied in kit form, accompanied by instructions for its installation, manufacturers should provide suitable information regarding temperature limitations in the installation instructions.

247. SECTION 23.1045 (as amended by amendment 23-7) COOLING TEST PROCEDURES FOR TURBINE ENGINE-POWERED AIRPLANES.

a. Explanation.

(1) Purpose. Cooling tests are conducted to determine the ability of the powerplant cooling provisions to maintain the temperatures of powerplant components and engine fluids within the temperature limits for which they have been certificated. These limits will normally be specified on the TC data sheet.

(2) Components With Time/Temperature Limits. The conventional method of approving engine components is to establish a temperature limit that will ensure satisfactory operation during the overhaul life of the engine. However, a component that exceeds the temperature limit can be approved at the elevated temperature for a specific period of time. To ensure that a component having a time/temperature limit will operate within the established limitation, a means should be provided to record the time and temperature of any excessive temperature and warn the pilot accordingly. The method of recording elapsed time and temperature should be automatic or activated by the pilot with a simple operation. Operating limitations requiring the pilot to detect a critical airplane operating condition and record the elapsed time in the airplane logs would not be acceptable due to the other pilot duties during the critical airplane operating condition.

(3) Altitude. Cooling tests should be conducted under critical ground and flight operating conditions to the maximum altitude for which approval is requested.

b. Test Procedures Applicable to Both Single-Engine and Multiengine Airplanes.

(1) Performance and Configuration. Refer to §§ 23.65 and 23.67, which have performance requirements related to engine cooling.

(2) Moisture. The tests should be conducted in air free of visible moisture.

(3) Weight and C.G. Forward c.g. at maximum gross weight is usually the most critical condition.

(4) Oil Quantity. The critical condition should be tested.

(5) Thermostat. Airplanes which incorporate a thermostat in the engine oil system may have the thermostat retained, removed, or blocked in such a manner as to pass all engine oil through the oil cooler. If the thermostat is retained, the oil temperature readings obtained on a cooler day corrected to hot-day conditions may therefore be greater than those obtained under actual hot-day conditions. Caution should be exercised when operating an airplane with the thermostat removed or blocked during cold weather to prevent failure of the lubricating system components.

(6) Instrumentation. Accurate and calibrated temperature-measuring devices should be used, along with acceptable thermocouples or temperature-pickup devices. The proper pickup should be located at critical engine positions.

(7) Generator. The alternator/generator should be electrically loaded to the rated capacity for the engine/accessory cooling tests.

(8) Temperature Limitations. For cooling tests, a maximum anticipated temperature (hot-day conditions) of at least 100°F at sea level must be used. Temperatures at higher altitudes assume a change at 3.6°F per 1000 feet of altitude, up to -69.7°F. The maximum ambient temperature selected and demonstrated satisfactorily becomes an airplane operating limitation per the requirements of § 23.1521(e).

(9) Temperature Stabilization. For the cooling tests, a temperature is considered stabilized when its observed rate of change is less than 2°F per minute.

(10) Altitude. The cooling tests should be started at the lowest practical altitude, usually below 3000 feet MSL, to provide a test data point reasonably close to sea level.

(11) Temperature Correction for Ground Operation. Recorded ground temperatures should be corrected to the maximum ambient temperature selected, without consideration of the altitude temperature lapse rate. For example, if an auxiliary power unit is being tested for ground cooling margins, the cooling margin should be determined from the recorded ground temperature, without regard to the test site altitude.

c. Test Procedures for Single-Engine, Turbine-Powered Airplanes.

(1) A normal engine start should be made and all systems checked out. The engine should be run at ground idle and temperatures and other pertinent data should be recorded.

(2) Taxi airplane for approximately 1 mile to simulate normal taxi operations. Record cooling data at 1-minute intervals.

(3) For hull-type seaplanes operating on water, taxi tests should be conducted such that spray characteristics do not bias the cooling characteristics. Engine cooling during water taxiing should be checked by taxiing downwind at a speed approximately 5 knots above the step speed for a minimum of 10 minutes continuous. Record cooling data at 1-minute intervals.

(4) Establish a pretakeoff holding condition on the taxiway (crosswind) for 20 minutes minimum or until temperatures stabilize. Record cooling data at 5-minute intervals.

(5) On the runway, set takeoff power and record cooling data.

(6) Takeoff as prescribed in § 23.51 and climb to pattern altitude. Record cooling data upon reaching pattern altitude or at 1-minute intervals if it takes more than 1-minute to reach pattern altitude.

(7) Retract flaps and continue climb with maximum continuous power at the speed selected to meet the requirements of § 23.65(c). Climb to the maximum approved altitude, recording cooling data at 1-minute intervals.

(8) Cruise at maximum continuous power (or V_{MO}/M_{MO}, if limiting) at maximum operating altitude until temperatures stabilize. Record cooling data at 1-minute intervals. For many components, this will be the critical temperature operating condition.

(9) Conduct a normal descent at V_{MO}/M_{MO} to holding altitude and hold until temperatures stabilize. Record cooling data at 1-minute intervals.

(10) Conduct a normal approach to landing. Record cooling data at 1-minute intervals.

(11) From not less than 200 feet above the ground, perform a balked landing go-around in accordance with § 23.77. Record cooling data at 1-minute intervals during a traffic pattern circuit.

(12) Climb to pattern altitude, perform a normal approach and landing in accordance with the applicable portion of § 23.75. Record cooling data at 1-minute intervals.

(13) Taxi back to ramp. Shut down engines. Allow engine to heat-soak. Record temperature data at 1-minute intervals until 5 minutes after temperatures peak.

d. Test Procedures for Multiengine, Turbine-Powered Airplanes. A multiengine airplane should conduct the same profile as the single-engine airplane, in an all-engine configuration. On completion of the all-engine profile, conduct the applicable one-engine-inoperative cooling climb test recording data at 1-minute intervals. Shut down critical engine and with its propeller (if applicable) in the minimum drag position, the remaining engine(s) at not more than maximum continuous power, or thrust, landing gear retracted, and wing flaps in the most favorable position. Climb at the speed used to show compliance with § 23.67. Continue until 5 minutes after temperatures peak.

e. Data Acquisition. The following data should be recorded at the time intervals specified in the particular test program. The data may be manually recorded unless the quantity and frequency necessitate automatic or semiautomatic means:

(1) Outside air temperature (OAT).

(2) Altitude.

(3) Airspeed (knots).

(4) Gas generator r.p.m.

(5) Engine torque.

(6) Time.

(7) Propeller r.p.m.

(8) Engine oil temperature.

(9) Pertinent engine temperature.

(10) Pertinent nacelle and component temperatures.

f. Data Reduction.

(1) Limitations. A maximum anticipated temperature (hot-day conditions) of at least $100^{\circ}F$ at sea level must be used. The assumed temperature lapse rate is $3.6^{\circ}F$ (or $2^{\circ}C$) per 1000 feet altitude up to the altitude at which a temperature of $-69.7^{\circ}F$ is reached, above which altitude the temperature is constant at $-69.7^{\circ}F$. On turbine engine-powered airplanes, the maximum ambient temperature selected becomes an airplane operating limitation in accordance with the requirements of § 23.1521(e). On turbine-powered airplanes, the applicant should correct the engine temperatures to as high a value as possible in order to not be limited.

(2) Correction Factors. Unless a more rational method applies, a correction factor of 1.0 is applied to the temperature data as follows: corrected temperature = true temperature + 1.0 [100 - 0.0036 (Hp) - true OAT].

Sample Calculation	
True Temperature	$300^{\circ}F$
True OAT	$15^{\circ}F$
Hp	5000 ft.

The corrected temperature = 300 + 1.0 [100 - 0.0036 (5000) - 15] = $367^{\circ}F$.

The corrected temperature is then compared with the maximum permissible temperature to determine compliance with the cooling requirements.

248. SECTION 23.1047 (as amended by amendment 23-21) COOLING TEST PROCEDURES FOR RECIPROCATING ENGINE-POWERED AIRPLANES.

a. Procedures.

(1) Additional Procedures. The procedures of paragraph 247b(1) through 247b(6) of this AC also apply to reciprocating engines.

(2) Altitude. Engine cooling tests for reciprocating engine airplanes are normally initiated below 2000 feet pressure altitude. Service experience indicates that engine cooling tests started above 5000 feet may not assure adequate cooling margins when the airplane is operated at sea level. If an applicant elects not to take the airplane to a low altitude test site, additional cooling margins have been found acceptable. If engine cooling tests cannot be commenced below 2000 feet pressure altitude, the temperature margin should be increased by 30°F at 7000 feet for cylinder heads and 60°F for both engine oil and cylinder barrels with a straight line variation from sea level to 7000 feet unless the applicant demonstrates that some other correction margin is more applicable.

(3) Hull-Type Seaplanes. Cooling tests on hull-type seaplanes should include, after temperatures stabilize, a downwind taxi for 10 minutes at 5 knots above the step speed, recording cooling data at 1-minute intervals.

(4) Test Termination. If at any time during the test, temperatures exceed the manufacturer's specified limits, the test is to be terminated.

(5) Climb Transition. At the beginning of the cooling climb, caution should be used in depleting the kinetic energy of the airplane while establishing the climb speed. The climb should not be started by "zooming" into the climb. The power may be momentarily reduced provided that the stabilized temperatures are not allowed to drop excessively. This means that a minimum of time should be used in slowing the airplane from the high cruise speed to the selected cooling climb speed. This may be accomplished by maneuver loading the airplane or any other means that provide minimum slow-down time.

(6) Component Cooling. Accessories or components on the engine or in the engine compartment which have temperature limits should be tested and should be at their maximum anticipated operating conditions during the cooling tests; for example, generators should be at maximum anticipated loads.

(7) Superchargers. Superchargers and turbosuperchargers should be used as described in the AFM. Engine cooling should be evaluated in the cruise condition at the maximum operating altitude, since this may be a more critical point than in climb. Also, turbocharged engines sometimes give a false peak and the climb should be continued long enough to be sure that the temperatures do not begin to increase again.

(8) Single-Engine Airplanes. The cooling tests for single-engine airplanes should be conducted as follows:

(i) At the lowest practical altitude, establish a level flight condition at not less than 75% maximum continuous power until temperatures stabilize. Record cooling data.

(ii) Increase engine power to takeoff rating and climb at a speed not greater than V_y except that a speed higher than V_y can be used under certain conditions. If a speed higher than V_y is chosen, § 23.1047(b) requires that the slope requirements of § 23.65 be met and a cylinder head temperature indicator is required, as specified in § 23.1305(f). Maintain takeoff power for 1 minute. Record cooling data.

(iii) At the end of 1 minute, reduce engine power to maximum continuous and continue climb for at least 5 minutes after temperatures peak or the maximum operating altitude is reached. Record cooling data at 1-minute intervals. If a leaning schedule is furnished to the pilot, it should be used.

(9) Multiengine Airplanes. For multiengine-powered airplanes that meet the minimum one-engine-inoperative climb performance specified in §§ 23.67(a) or 23.67(b)(1) with the airplane in the configuration used in establishing critical one-engine-inoperative climb performance:

(i) At the lower altitude of 1000 feet below engine critical altitude or 1000 feet below the altitude at which the one-engine-inoperative rate of climb is 0.027 V_{so}^2, or at the lowest practical altitude (when applicable), stabilize temperatures of the test engine in level flight at not less than 75% maximum continuous power. Record cooling data.

(ii) After temperatures stabilize, initiate a climb at a speed not more than the highest speed at which compliance with the climb requirement of §§ 23.67(a) or 23.67(b)(1) is shown. If the speed exceeds V_{YSE}, § 23.1047(b)(2) requires a cylinder head temperature indicator be provided. With the test engine at maximum continuous power (or full throttle), continue climb until 5 minutes after temperatures peak or the maximum operating altitude is reached. Record cooling data at 1-minute intervals.

(10) Performance Limited Multiengine Airplanes. For multiengine airplanes that cannot meet the minimum one-engine-inoperative performance specified in §§ 23.67(a) or 23.67(b)(1):

(i) Set zero thrust on the planned "inoperative" engine and determine an approximate rate of sink (or climb). A minimum safe test altitude should then be established.

(ii) Stabilize temperatures in level flight with engines operating at no less than 75% maximum continuous power and as near sea level as practicable or the minimum safe test altitude.

(iii) After temperatures stabilize, initiate climb at best rate-of-climb speed or minimum rate-of-descent speed, as applicable, with one engine inoperative and remaining engine(s) at maximum continuous power. Continue for at least 5 minutes after temperatures peak. Record cooling data at 1-minute intervals.

b. Data Acquisition. The following data should be recorded at the time intervals specified in the applicable test programs and may be manually recorded unless the quantity and frequency necessitate automatic or semiautomatic means:

(1) Time.

(2) Hottest cylinder head temperature.

(3) Hottest cylinder barrel temperature (only if a limitation).

(4) Engine oil inlet temperature.

(5) Outside air temperature.

(6) Indicated airspeed (knots).

(7) Pressure altitude.

(8) Engine r.p.m.

(9) Propeller r.p.m.

(10) Manifold pressure.

(11) Carburetor air temperature.

(12) Mixture setting.

(13) Throttle setting.

(14) Temperatures of components or accessories which have established limits that may be affected by powerplant heat generation.

c. <u>To Correct Cylinder Barrel Temperature to Anticipated Hot-Day Conditions</u>.

(1) Corrected cylinder barrel temperature = true observed cylinder barrel temperature + 0.7 [100 - 0.0036 (pressure altitude) - true OAT].

(2) For example:

True observed maximum cylinder barrel temperature	244°F.
Pressure Altitude	8330 ft.
True OAT	+55°F

(3) Corrected cylinder barrel temperature = 244 + 0.7 [100 - 0.0036 (8330) - 55] = 255°F.

(4) The corrected temperatures are then compared with the maximum permissible temperatures to determine compliance with cooling requirements.

d. <u>To Correct Cylinder Head or Other Temperatures to Anticipated Hot-Day Conditions</u>.

(1) Corrected temperature = true temperature + 1.0 [100 - 0.0036 (pressure altitude) - true outside air temperature].

(2) For example:

True maximum cylinder head temperature	406°F.
Pressure Altitude	8330 ft.
True OAT	+55°F

(3) Corrected cylinder head temperature = 406 + 1.0 [100 - 0.0036 (8330) - 55] = 421°F.

(4) The corrected temperatures are then compared with the maximum permissible temperatures to determine compliance with cooling requirements.

e. Liquid Cooled Engines. (RESERVED).

249.-254. RESERVED.

Section 6. INDUCTION SYSTEM

255. SECTION 23.1091 (as amended by amendment 23-7) AIR INDUCTION. AC 20-124 covers the turbine engine water ingestion aspects of this requirement.

256. SECTION 23.1093 (as amended by amendment 23-29) INDUCTION SYSTEM ICING PROTECTION.

a. Explanation.

(1) Purpose. Tests of engine induction system icing protection provisions are conducted to ensure that the engine is able to operate throughout its flight power range without adverse effect on engine operation. Reciprocating engines utilize a preheater or a sheltered alternate air source to provide adequate heat rise to prevent or eliminate ice formation in the engine induction system. The adequacy of this heat rise is evaluated during the test. The amount of heat available is determined by measuring the intake heat rise by temperature measurements of the air before it enters the carburetor. Turbine engine inlet ducts must be protected to prevent the accumulation of ice as specified in § 23.1093(b)(1).

(2) Reciprocating Sea Level Engine Configurations.

(i) Venturi Carburetor. Section 23.1093(a)(1) requires a 90°F heat rise at 75% maximum continuous power at 30°F OAT.

(ii) Single-Engine Airplanes With a Carburetor Tending to Prevent Icing (Pressure Carburetor). Section 23.1093(a)(4) requires an alternate air source with a temperature equal to that of the air downstream of the cylinders.

(iii) Multiengine Airplane With Carburetors Tending to Prevent Icing (Pressure Carburetor). Section 23.1093(a)(5) requires a 90°F heat rise at 75% maximum continuous power at 30°F OAT.

(iv) Fuel Injection With Ram Air Tubes. A heat rise of 90°F at 75% maximum continuous power is recommended.

(v) Fuel Injection Without Projections Into the Induction Air Flow. An alternate air source with a temperature not less than the cylinder downstream air is recommended.

(3) Reciprocating Altitude Engine Configurations.

(i) Venturi Carburetor. Section 23.1093(a)(2) requires a 120°F heat rise at 75% maximum continuous power at 30°F OAT.

(ii) Carburetors Tending to Prevent Icing (Pressure Carburetor). Section 23.1093(a)(3) requires a heat rise of 100°F at 60% maximum continuous power at 30°F OAT or 40°F heat rise if an approved fluid deicing system is used.

(iii) Fuel Injection. Same as for sea level fuel injected engines.

(4) Turbine Engines. Section 23.1093(b) requires turbine engines to be capable of operating without adverse effects on operation or serious loss of power or thrust under the icing conditions specified in Part 25, appendix C. The powerplant should be protected from ice at all times, whether or not the airplane is certificated for flight into known icing conditions.

b. Reciprocating Engine Test Considerations.

(1) Visible Moisture. The tests should be conducted in air free of visible moisture.

(2) Instrumentation. All instruments used during the test should be calibrated and all calibration curves made part of the Type Inspection Report. Calibrations should be made of complete systems as installed and shall cover the temperature range expected during the tests.

(3) Heat Rise. All carburetor air heat rise requirements should be met at an outside air temperature of 30°F. If the test cannot be conducted in an atmosphere with an ambient air temperature of 30°F, it will normally be flown at low, intermediate, and high altitudes. If a 30°F day exists at an altitude where 75% of rated power is available, only one test is necessary.

(4) Intake Air. Care should be exercised to assure that the method of measuring the temperature of the air will give an indication of the average temperature of the airflow through the intake and not just a stratum of air. This may be accomplished by temperature measurements of the intake air at several points. Usually, the temperature probe is placed at the carburetor deck; however, test data may be obtained with the pickup at other locations. A carburetor throat temperature pickup in lieu of carburetor air box temperature instrumentation will not suffice for accurate readings unless calibration data is made available to correlate carburetor throat temperatures to actual air inlet temperatures.

c. Test Procedures for Reciprocating Engine Airplanes.

(1) At low altitude, stabilize airplane with full throttle or, if the engine is supercharged, with maximum continuous power on the test engine. With carburetor air heat control in the "cold" position record data. Manually operated turbochargers should be off. For integrally turbocharged engines, heat rise data should be taken at lowest altitude conditions, where the turbo is providing minimum output.

(2) Apply carburetor heat and after condition stabilizes, record data.

(3) Reduce airspeed to 90% of that attained under item (1). With carburetor air heat control in the "cold" position and condition stabilized, record data.

(4) Apply carburetor heat and after condition stabilizes, record data.

(5) Reduce airspeed to 80% of that attained under item (1). With carburetor air heat control in the "cold" position and condition stabilized, record data.

(6) Apply carburetor heat and after condition stabilizes, record data.

(7) At the intermediate altitude, repeat steps (1) through (6).

(8) At high altitude, repeat steps (1) through (6). Data to be recorded.

(i) Altitude (feet).

(ii) Airspeed (IAS) (Knots).

(iii) Ambient air temperature ^{o}F.

(iv) Carburetor air temperature ^{o}F.

(v) Carburetor heat control position.

(vi) Engine r.p.m.

(vii) Engine manifold pressure (in Hg).

(viii) Throttle position.

d. Data Reduction. Figures 256-1 and 256-2 show sample carburetor air heat rise determinations.

e. Test Procedures for Turbine Engine-Powered Airplanes. Tests to determine the capability of the turbine engine to operate throughout its flight power range without adverse effect on engine operation or serious loss of power or thrust

NOTE: May be flown at only one altitude if O.A.T. of 30°F is Available	MINIMUM ALTITUDE						INTERMEDIATE ALTITUDE						MAXIMUM ALTITUDE (75%)					
	Full Throttle or MC Power*		90% IAS of Column #1		80% IAS of Column #1		Full Throttle of MC Power*		90% IAS of Column #1		80% IAS of Column #1		Full Throttle or MC Power*		90% IAS of Columnn #1		80% IAS of Column #1	
Carburetor Air Heat Control Position	C	H	C	H	C	H	C	H	C	H	C	H	C	H	C	H	C	H
Pressure Altitude (ft.)	1500						5000						8000					
O.A.T. (F)	83	83	83	83	83	83	72	72	72	72	72	72	60	60	60	60	60	60
C.A.T. (F)	84	215	84	205	84	200	73	201	73	189	73	184	61	190	61	185	61	176
Heat Rise		132		122		117		129		117		112		130		125		116
I.A.S. (M.P.H.)	105	99	95	92	84	82	96	88	87	78	77	70	90	80	82	75	72	67
R.P.M.	2850	2730	2690	2590	2430	2310	2800	2640	2555	2400	2410	2280	2770	2525	2665	2480	2525	2310
M.P. (In. Hg.)	26.4	25.7	24.0	23.5	22.0	21.3	23.5	22.8	19.6	19.3	19.0	18.5	21.2	20.4	19.9	19.4	18.0	17.2
Indicated B.H.P.	144	132	120	112	105	99	125	114	92	85	76	72	113	100	101	90	73	65
Std. Temperature for Pressure Altitude (F)	54						41						30					
Temperature Correction Factor (See note 1)	.972	.872	.972	.879	.972	.882	.970	.870	.970	.879	.970	.882	.970	.868	.970	.871	.970	.878
Actual B.H.P.	140	115	117	98.4	102	87.4	121	99.2	89	74.7	74	63.5	110	86.8	98	78.4	71	57.1
% Rated B.H.P. (See note 2)	100	82.2	83.5	70.2	72.8	62.4	86.4	71.0	63.5	53.4	52.8	45.3	78.5	62.1	70	56.0	50.6	40.8
Throttle Position	FT	FT	P	P	P	P	FT	FT	P	P	P	P	FT	FT	P	P	P	P

*Supercharged Engines Only

NOTE 1: Temperature Correction Factor = $\sqrt{\dfrac{\text{std temp } (^{o}F) + 460}{\text{CAT } (^{o}F) + 460}}$

NOTE 2: Rated BHP = 140

NOTE 3: Circled numbers indicate data plotted on figure 256-2.

Figure 256-1 - CARBURETOR AIR HEAT RISE CALCULATIONS

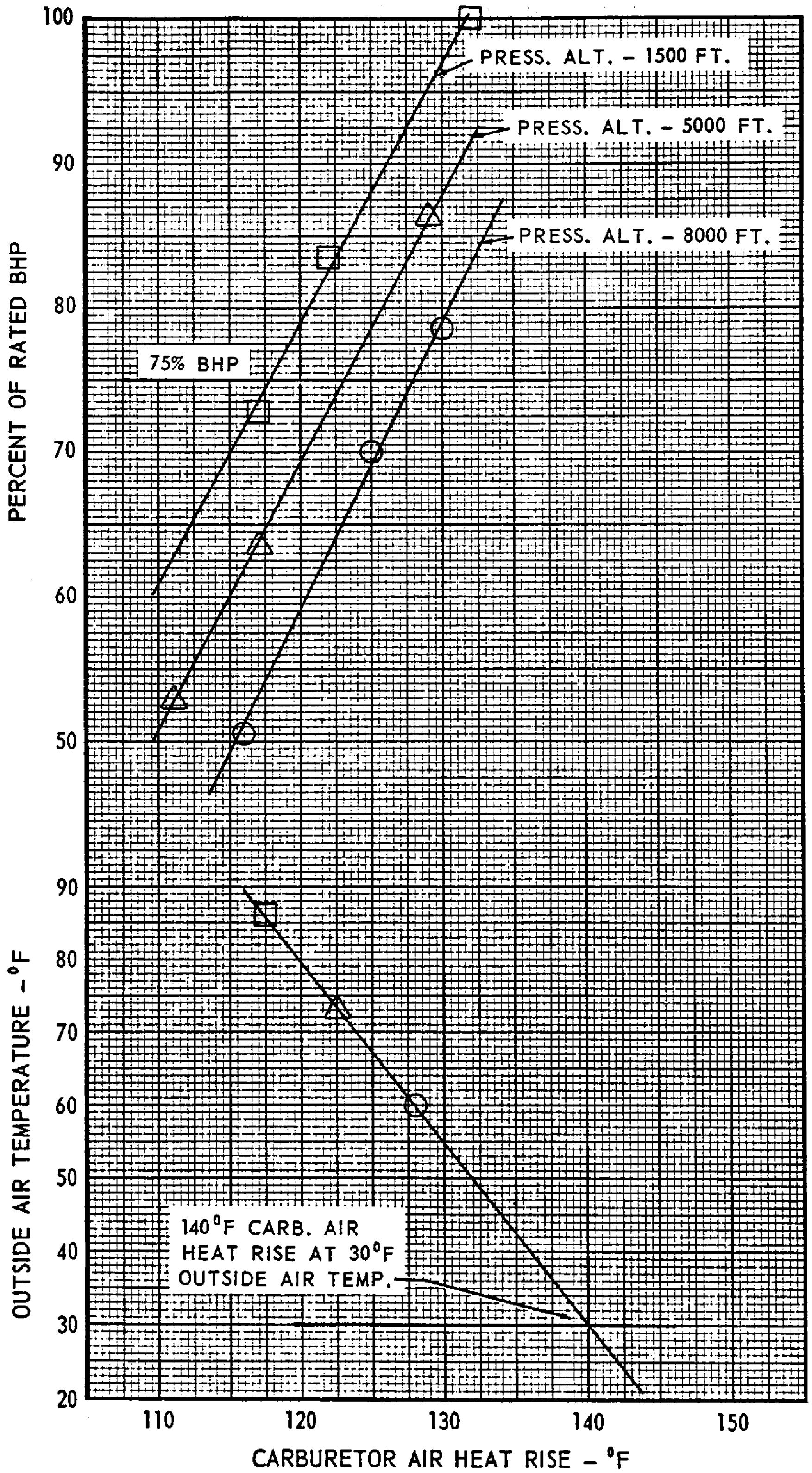

Figure 256-2 - CARBURETOR AIR HEAT RISE PLOTS should be conducted to encompass the icing conditions specified in Part 25, appendix C. Each airplane should be evaluated for compliance. Thermodynamic exercises and dry air tests alone are not usually adequate, and actual icing encounters or wind tunnel testing are necessary.

257.-265. RESERVED.

Section 7. POWERPLANT CONTROLS AND ACCESSORIES

266. SECTION 23.1141 (as amended by amendment 23-18) POWERPLANT CONTROLS: GENERAL.

a. Explanation. Powerplant controls for each powerplant function will be grouped for each engine allowing simultaneous or independent operation as desired. Each control will be clearly marked as to function and control position. (Also see § 23.777). Controls are required to maintain any position set by the pilot without tendency to creep due to vibration or control loads.

b. Procedures. None.

267. SECTION 23.1145 (as amended by amendment 23-18) IGNITION SWITCHES. (RESERVED).

268. SECTION 23.1153 (original issue) PROPELLER FEATHERING CONTROLS.

a. Explanation. If the propeller pitch or speed control lever also controls the propeller feathering control, some means are required to prevent inadvertent movement to the feathering position.

b. Procedures. None.

269.-278. RESERVED.

Section 8. POWERPLANT FIRE PROTECTION

279. SECTION 23.1189 (as amended by amendment 23-29) SHUTOFF MEANS.

a. Explanation. The location and operation of any required shutoff means is substantiated by analysis of design data, inspection, or test. The location and guarding of the control (switch) and the location and clarity of any required indicators should be evaluated.

b. Procedures. Control locations and guarding and indicator effectiveness should be part of the complete cockpit evaluation. Check the shutoff means function by performing an after-flight engine shutdown using the fuel shutoff.

280.-285. RESERVED.

CHAPTER 5. EQUIPMENT
Section 1. GENERAL

286. SECTION 23.1301 (prior to amendment 23-20) FUNCTION AND INSTALLATION. A system that is not essential for safe operation, nor required by regulation, may be approved if it is not a hazard in normal operation or when it malfunctions or fails. It does not have to perform its intended function, but must be evaluated for hazards. An example of such a system is an engine fire extinguisher system. The operating information should include a placard and an entry in the AFM to the effect that the capability of the fire extinguisher system to perform its intended function has not been evaluated by the FAA.

287. SECTION 23.1301 (as amended by amendment 23-20) FUNCTION AND INSTALLATION.

a. Explanation. Section 23.1301 gives specific installation requirements. Particular attention should be given to those installations where an external piece of equipment could affect the flight characteristics. All installations of this nature should be evaluated by the flight test pilot to verify that the equipment functions properly when installed.

b. Avionics Test.

(1) Very High Frequency (VHF) Communication Systems. See AC 20-67B. AC 20-67B references Radio Technical Commission for Aeronautics (RTCA) document DO-186. DO-186, paragraph 3.4.2.3 speaks to ground facility coverage area. FAA Order 6050.32, appendix 2, shows the coverage limits for various facility parameters. Contact the nearest FAA Airway Facilities Sector Office to examine the order.

(2) High Frequency (HF) Communication Systems.

(i) Ground Station Contacts. Acceptable communication should be demonstrated by contacting a ground station on as wide a range of frequencies as HF propagation conditions allow. Distances may vary from 100 to several hundred nautical miles (n.m.). The system should perform satisfactorily in its design modes.

(ii) Precipitation Static. It should be demonstrated that precipitation static is not excessive when the airplane is flying at cruise speed (in areas of high electrical activity, including clouds and rain if possible). Use the minimum amount of installed dischargers for which approval is sought.

(iii) Electromagnetic Compatibility (EMC). Electromagnetic compatibility tests should be conducted on the ground and in flight at 1.0 MHz intervals. Any electromagnetic interference (EMI) noted on the ground should be repeated in flight at the frequency at which the EMI occurred on the ground. Since squat switches may isolate some systems from operation on the ground (i.e., air data system, pressurization, etc.), EMI should be evaluated with all systems operating in flight to verify that no adverse effects are present in the engine, fuel control computer, brake antiskid, etc., systems.

(3) Very High Frequency Omnirange (VOR) Systems.

(i) Antenna Radiation Patterns. These flight tests may be reduced if adequate antenna radiation pattern studies have been made and these studies show the patterns to be without significant holes (with the airplane configurations used in flight; that is, flaps, landing gear, etc.). Particular note should be made in recognition that certain propeller r.p.m. settings may cause modulation of the course deviation indication (prop-modulation). This information should be made a part of the AFM.

(A) Reception. The airborne VOR system should operate normally with warning flags out of view at all headings of the airplane (wings level) throughout the standard service volumes depicted in the Airman's Information Manual (AIM) up to the maximum altitude for which the airplane is certified.

(B) Accuracy. The accuracy determination should be made such that the indicated reciprocal agrees within 2°. Tests should be conducted over at least two known points on the ground such that data are obtained in each quadrant. Data should correlate with the ground calibration and in no case should the absolute error exceed $\pm 6^{\circ}$. There should be no excessive fluctuation in the course deviation indications.

(ii) En Route Reception. Fly from a VOR facility rated for high altitude along a radial at an altitude of 90% of the airplane's maximum certificated altitude to the standard service volume range. The VOR warning flag should not come into view, nor should there be deterioration of the station identification signal. The course width should be $20^{\circ} \pm 5^{\circ}$ (10° either side at the selected radial). The tests should be flown along published route segments to preclude ground station anomalies. If practical, perform an en route segment on a doppler VOR station to verify the compatibility of the airborne unit. Large errors have been found when incompatibility exists. Contact the nearest FAA Airway Facilities Sector Office to locate a doppler VOR.

(iii) Low-Angle Reception. Perform a 360° right and 360° left turn at a bank angle of at least 10° at an altitude just above the lower edge of the standard service volume and at the maximum service volume distance. Signal dropout should not occur as evidenced by the warning flag appearance. Dropouts that are relieved by a reduction of bank angle at the same relative heading to the station are satisfactory. The VOR identification should be satisfactory during the left and right turns.

(iv) High-Angle Reception. Repeat the turns described in (iii) above, but at a distance of 50-70 n.m. (20-30 n.m. for airplanes not to be operated above 18,000 feet) from the VOR facility and at an altitude of at least 90% of the maximum certificated altitude of the airplane.

(v) En Route Station Passage. Verify that the to-from indicator correctly changes as the airplane passes through the cone of confusion above a VOR facility.

(vi) VOR Approach. Conduct VOR approaches with gear and flaps down. With the facility 12-15 n.m. behind the airplane, use sufficient maneuvering in the approach to ensure the signal reception is maintained during beam tracking.

(vii) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify, by observation, that no adverse effects are present in the required flight systems.

(4) Localizer Systems.

(i) Antenna Radiation Patterns. Flight test requirements should be modified to allow for adequate antenna radiation pattern measurements as discussed in VOR systems, subparagraph (3)(i).

(A) Signal Strength. The signal input to the receiver, presented by the antenna system, should be of sufficient strength to keep the malfunction indicator out of view when the airplane is in the approach configuration (landing gear extended - approach flaps) and within the normal limits of localizer coverage shown in the Airman's Information Manual (AIM). This signal should be received for 360° of airplane heading at all bank angles up to 10° left or right at all normal pitch attitudes and at an altitude of approximately 2000 feet (see RTCA Document DO-102).

(B) Bank Angles. Satisfactory results should also be obtained at bank angles up to 30° when the airplane heading is within 60° of the inbound localizer course. Satisfactory results should result with bank angles up to 15° on headings from 60° to 90° of the localizer inbound course and up to 10° bank angle on headings from 90° to 180° from the localizer inbound course.

(C) Course Deviation Indicator (CDI). The deviation indicator should properly direct the airplane back to course when the airplane is right or left of course.

(D) Station Identification. The station identification signal should be of adequate strength and sufficiently free from interference to provide positive station identification, and voice signals should be intelligible with all electric equipment operating and pulse equipment transmitting.

(ii) Localizer Intercept. In the approach configuration and at a distance of at least 18 n.m. from the localizer facility, fly toward the localizer front course, inbound, at an angle of at least 50°. Perform this maneuver from both left and right of the localizer beam. No flags should appear during the time the deviation indicator moves from full deflection to on-course.

(iii) Localizer Tracking. While flying the localizer inbound and not more than 5 miles before reaching the outer marker, change the heading of the airplane to obtain full needle deflection. Then fly the airplane to establish localizer on-course operation. The localizer deviation indicators should direct the airplane to the localizer on-course. Perform this maneuver with both a left and a right needle deflection. Continue tracking the localizer until over the transmitter. Acceptable front course and back course approaches should be conducted to 200 feet or published minimums.

(iv) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify, by observation, that no adverse effects are present in the required flight system.

(5) Glide Slope Systems.

(i) Signal Strength. The signal input to the receiver should be of sufficient strength to keep the warning flags out of view at all distances to 10 n.m. from the facility. This performance should be demonstrated at all airplane headings between 30° right and left of the localizer course (see RTCA Document DO-1010). The deviation indicator should properly direct the airplane back to path when the airplane is above or below path. Interference with the navigation operation, within 10 n.m. of the facility, should not occur with all airplane equipment operating and all pulse equipment transmitting. There should be no interference with other equipment as a result of glide slope operation.

(ii) Glide Slope Tracking. While tracking the glide slope, maneuver the airplane through normal pitch and roll attitudes. The glide slope deviation indicator should show proper operation with no flags. Acceptable approaches to 200 feet or less above threshold should be conducted.

(iii) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify, by observation, that no adverse effects are present in the required flight systems.

(6) Marker Beacon System.

(i) Flight test.

(A) In low sensitivity, the marker beacon annunciator light should be illuminated for a distance of 2000 to 3000 feet when flying at an altitude of 1000 feet AGL on the localizer centerline in all flap and gear configurations.

(B) An acceptable test to determine distances of 2000 to 3000 feet is to fly at a ground speed listed in table 1 and time the marker beacon light duration.

Table 1 - LIGHT DURATION

Altitude = 1000 feet (AGL)

Ground Speed	Light Time (Seconds)	
Knots	2000 feet	3000 feet
90	13	20
110	11	16
130	9	14
150	8	12

(C) For ground speeds other than tabled values, the following formulas may be used:

$$\text{Upper limit (seconds)} = \frac{1775}{\text{Ground Speed in Knots}}$$

$$\text{Lower limit (seconds)} = \frac{1183}{\text{Ground Speed in Knots}}$$

(D) In high sensitivity, the marker beacon annunciator light and audio will remain on longer than when in low sensitivity.

(E) The audio signal should be of adequate strength and sufficiently free from interference to provide positive identification.

(F) As an alternate procedure, cross the outer marker at normal ILS approach altitudes and determine adequate marker aural and visual indication.

(ii) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify, by observation, that no adverse effects are present in the required flight system.

(7) Automatic Direction Finding (ADF) Equipment.

(i) Range and Accuracy. The ADF system installed in the airplane should provide operation with errors not exceeding 5°, and the aural signal should be clearly audible up to the distance listed for any one of the following types of radio beacons:

(A) 75 n.m. from an HH facility.

(B) 50 n.m. from an H facility. Caution - service ranges of individual facilities may be less than 50 n.m.

(C) 25 n.m. from an MH facility.

(D) 15 n.m. from a compass locator.

(ii) Needle Reversal. The ADF indicator needle should make only one 180° reversal when the airplane flies over a radio beacon. This test should be made with and without the landing gear extended.

(iii) Indicator Response. When switching stations with relative bearings differing by $180^\circ \pm 5^\circ$, the indicator should indicate the new bearing within $\pm 5^\circ$ in not more than 10 seconds.

(iv) Antenna Mutual Interaction. For dual installations, there should not be excessive coupling between the antennas.

(v) Technique.

(A) Range and Accuracy. Tune in a number of radio beacons spaced throughout the 190-535 KHz range and located at distances near the maximum range for the beacon. The identification signals should be understandable and the ADF should indicate the approximate direction to the stations. Beginning at a distance of at least 15 n.m. from a compass locator in the approach configuration (landing gear extended, approach flaps), fly inbound on the localizer front course and make a normal ILS approach. Evaluate the aural identification signal for strength and clarity and the ADF for proper performance with the receiver in the ADF mode. All electrical equipment on the airplane should be operating and all pulse equipment should be transmitting. Fly over a ground or appropriately established checkpoint with relative bearings to the facility of 0°, 45°, 90°, 135°, 180°, 225°, 270°, and 315°. The indicated bearings to the station should correlate within 5°. The effects of the landing gear on bearing accuracy should be determined. (A calibration placard should be provided, if appropriate.)

(B) Needle Reversal. Fly the airplane over an H, MH, or compass locator facility at an altitude 1000 to 2000 feet above ground level. Partial reversals which lead or lag the main reversal are permissible.

(C) Indicator Response. With the ADF indicating station dead ahead, switch to a station having a relative bearing of 175°. The indicator should indicate within $\pm 5^\circ$ of the bearing in not more than 10 seconds.

(D) Antenna Mutual Interaction.

(1) If the ADF installation being tested is dual, check for coupling between the antenna by using the following procedure.

(2) With #1 ADF receiver tuned to a station near the low end of the ADF band, tune the #2 receiver slowly throughout the frequency range of all bands and determine whether the #1 ADF indicator is adversely affected.

(3) Repeat (2) with the #1 ADF receiver tuned to a station near the high end of the ADF band.

(vi) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify, by observation, that no adverse effects are present in the required flight systems.

(8) Distance Measuring Equipment (DME).

(i) Tracking Performances. The DME system should continue to track without dropouts when the airplane is maneuvered throughout the airspace within the standard service volume of the VORTAC/DME station and at altitudes above the lower edge of the standard service volume to the maximum operating altitude. This tracking standard should be met with the airplane:

(A) In cruise configuration.

(B) At bank angle up to 10°.

(C) Climbing and descending at normal maximum climb and descent attitude.

(D) Orbiting a DME facility.

(E) Provide clearly readable identification of the DME facility.

(ii) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify, by observation, that no adverse effects are present in the required flight systems.

(iii) Climb and Maximum Distance. Determine that there is no mutual interference between the DME system and other equipment aboard the airplane. Beginning at a distance of at least 10 n.m. from a DME facility and at an altitude of 2000 feet above the DME facility, fly the airplane on a heading so that the airplane will pass over the facility. At a distance of 5-10 n.m. beyond the DME facility, operate the airplane at its normal maximum climb attitude up to 90% of the maximum operating altitude, maintaining the airplane on a station radial (within 5°). The DME should continue to track with no unlocks to the range of the standard service volume.

(iv) Long-Range Reception.

(A) Perform two 360° turns, one to the right and one to the left, at a bank angle of at least 10° at the maximum service volume distance of the DME facility and at an altitude of at least 90% of the maximum operating altitude.

(B) Unlocks may occur and are acceptable if they do not interfere with the intended flight path of the airplane or are relieved by a reduction of bank angle at the same relative heading to the station.

(v) High-Angle Reception. Repeat the flight pattern and observations of (iii) above at a distance of 50-70 n.m. (20-30 n.m. for airplanes not to be operated above 18,000 feet) from the DME facility and at an altitude of at least 90% of the maximum operating altitude.

(vi) Penetration. From 90% of the maximum operating altitude, perform a letdown directly toward the ground station using normal maximum rate of descent procedures to a DME facility so as to reach an altitude of 5000 feet above the DME facility 5-10 n.m. before reaching the DME facility. The DME should continue to track during the maneuver with no unlocks.

(vii) Orbiting. At an altitude of 2000 feet above the terrain, at holding pattern speeds appropriate for the type of airplane and with the landing gear extended, fly at least 15° sectors of left and right 35 n.m. orbital patterns around the DME facility. The DME should continue to track with no more than one unlock, not to exceed one search cycle, in any 5 miles of orbited flight.

(viii) Approach. Make a normal approach at an actual or simulated field with a DME. The DME should track without an unlock (station passage expected).

(ix) DME Hold. With the DME tracking, activate the DME hold function. Change the channel selector to a localizer frequency. The DME should continue to track on the original station.

(9) Transponder Equipment.

(i) Signal Strength. The ATC transponder system should furnish a strong and stable return signal to the interrogating radar facility when the airplane is flown in straight and level flight throughout the airspace within 160 n.m. of the radar station from radio line of sight to within 90% of the maximum altitude for which the airplane is certificated or to the maximum operating altitude. Airplanes to be operated at altitudes not exceeding 18,000 feet should meet the above requirements to only 80 n.m.

(ii) Single Site Tracking. Special arrangements should be made for single-site tracking. ATC coverage includes remote stations and unless single-site is utilized, the data may be invalid.

(iii) Dropout Times. When the airplane is flown within the airspace described above, the dropout time should not exceed 20 seconds in the following maneuvers:

(A) In turns at bank angles up to 10°.

(B) Climbing and descending at normal maximum climb and descent attitude.

(C) Orbiting a radar facility.

(iv) Climb and Distance Coverage.

(A) Beginning at a distance of at least 10 n.m. from and at an altitude of 2000 feet above that of the radar facility and using a transponder code assigned by the ARTCC, fly on a heading that will pass the airplane over the facility. Operate the airplane at its normal maximum climb attitude up to within 90% of the maximum altitude for which the airplane is certificated, maintaining the airplane at a heading within 5° from the radar facility. After reaching the maximum altitude for which the airplane is certificated, fly level at the maximum altitude to 160 (or 80) n.m. from the radar facility.

(B) Communicate with the ground radar personnel for evidence of transponder dropout. During the flight, check the "ident" mode of the ATC transponder to ensure that it is performing its intended function. Determine that the transponder system does not interfere with other systems aboard the airplane and that other equipment does not interfere with the operation of the transponder system. There should be no dropouts for two or more sweeps.

(v) Long-Range Reception. Perform two 360° turns, one to the right and one to the left, at bank angles of at least 10° with the flight pattern at least 160 (or 80) n.m. from the radar facility. During these turns, the radar display should be monitored and there should be no signal dropouts (two or more sweeps).

(vi) High-Angle Reception. Repeat the flight pattern and observations of (iv) above at a distance of 50 to 70 n.m. from the radar facility and at an altitude of at least 90% of the maximum operating altitude. There should be no dropout (two or more sweeps). Switch the transponder to a code not selected by the ground controller. The airplane secondary return should disappear from the scope. The controller should then change his control box to a common system and a single slash should appear on the scope at the airplane's position.

(vii) High-Altitude Cruise. Fly the airplane within 90% of its maximum certificated altitude or its maximum operating altitude beginning at a point 160 (or 80) n.m. from the radar facility on a course which will pass over the radar facility. There should be no transponder dropout (two or more sweeps) or "ring-around."

(viii) Holding and Orbiting Patterns.

(A) At an altitude of 2000 feet or minimum obstruction clearance altitude (whichever is greater) above the radar antenna and at holding pattern speeds, flaps and gear extended, fly one each standard rate 360° turn right and left at a distance of approximately 10 n.m. from the ARSR facility. There should be no signal dropout (two or more sweeps).

(B) At an altitude of 2000 feet or minimum obstruction clearance altitude (whichever is greater) above the radar antenna and at holding pattern speeds appropriate for the type of airplane, fly 45° sectors of left and right 10 n.m. orbital patterns around a radar facility with gear and flaps extended. There should be no signal dropout (two or more sweeps).

(ix) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify, by observation, that no adverse effects are present in the required flight systems.

(10) Weather Radar.

(i) Bearing Accuracy. The indicated bearing of objects shown on the display should be within $\pm 10^\circ$ of their actual relative bearing. Verify that as airplane turns to right or left of target, the indicated display moves in the opposite direction. Fly under conditions which allow visual identification of a target, such as an island, a river, or a lake, at a range of approximately 80% of the maximum range of the radar. When flying toward the target, select a course that will pass over a reference point from which the bearing to the target is known. When flying a course from the reference point to the target, determine the error in displayed bearing to the target on all range settings. Change heading in increments of 10° and determine the error in the displayed bearing to the target.

(ii) Distance of Operation. The radar should be capable of displaying distinct and identifiable targets throughout the angular range of the display and at approximately 80% of the maximum range.

(iii) Beam Tilting. The radar antenna should be installed so that its beam is adjustable to any position between 10° above and below the plane of rotation of the antenna. Tilt calibration should be verified.

(iv) Contour Display (Iso Echo).

(A) If heavy cloud formations or rainstorms are reported within a reasonable distance from the test base, select the contour display mode. The radar should differentiate between heavy and light precipitation.

(B) In the absence of the above weather conditions, determine the effectiveness of the contour display function by switching from normal to contour display while observing large objects of varying brightness on the indicator. The brightest object should become the darkest when switching from normal to contour mode.

(v) Antenna Stabilization, When Installed. While in level flight at 10,000 feet or higher, adjust the tilt approximately 2-3° above the point where ground return was eliminated. Roll right and left approximately 15°, then pitch down approximately 10° (or within design limits). No ground return should be present.

(vi) Ground Mapping. Fly over areas containing large, easily identifiable landmarks such as rivers, towns, islands, coastlines, etc. Compare the form of these objects on the indicator with their actual shape as visually observed from the cockpit.

(vii) Mutual Interference. Determine that no objectionable interference is present on the radar indicator from any electrical or radio/navigational equipment when operating and that the radar installation does not interfere with the operation of any of the airplane's radio/navigational systems.

(viii) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify, by observation, that no adverse effects are present in the required flight systems.

(ix) Light Conditions. The display should be evaluated during all lighting conditions, including night and direct sunlight.

(11) Area Navigation.

(i) Advisory Circular 90-45A. This AC is the basic criteria for evaluating an area navigation system, including acceptable means of compliance to the FAR.

(ii) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify, by observation, that no adverse effects are present in the required flight systems.

(12) Inertial Navigation.

(i) Basic Criteria. Advisory Circular 25-4 is the basic criteria for the engineering evaluation of an inertial navigation system (INS) and offers acceptable means of compliance with the applicable FAR. The engineering evaluation of an INS should also include an awareness of AC 121-13 which presents criteria to be met before an applicant can get operational approval. For flights up to

10 hours, the radial error should not exceed 2 n.m. per hour of operation on a 95% statistical basis. For flights longer than 10 hours, the error should not exceed ±20 n.m. cross-track or ±25 n.m. along-track error. A 2 n.m. radial error is represented by a circle, having a radius of 2 n.m., centered on the selected destination point.

(ii) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify, by observation, that no adverse effects are present in the required flight systems.

(13) Doppler Navigation.

(i) Doppler navigation system installed performance should be evaluated in accordance with AC 121-13.

(ii) Electromagnetic Compatibility (EMC). With all systems operating in flight, verify, by observation, that no adverse effects are present in the required flight systems.

(14) Audio Interphone Systems.

(i) Acceptable communications should be demonstrated for all audio equipment including microphones, speakers, headsets, and interphone amplifiers. All modes of operation should be tested, including operation during emergency conditions (that is, emergency descent, and oxygen masks) with all engines running, all pulse equipment transmitting and all electrical equipment operating. If aural warning systems are installed, they should be evaluated, including distinguishing aural warnings when using headphones and with high air noise levels.

(ii) Electromagnetic Compatibility (EMC). With all systems operating during flight, verify, by observation, that no adverse effects are present in the required flight systems.

(15) Electronic Flight Instrument Systems. This subject will be covered in a separate AC.

(16) VLF/Omega Navigation Systems. See AC's 20-101B, 90-79, 120-31A, and 120-37.

(17) LORAN C Navigation Systems. See AC 20-121.

(18) Microwave Landing Systems. (RESERVED).

288. SECTION 23.1303 (prior to amendment 23-17) FLIGHT AND NAVIGATION INSTRUMENTS.

a. Explanation. Section 23.1303, as amended by amendment 23-17, provides for a speed warning device for all turbine engine-powered airplanes. Section 23.1303(e) should not be applied to Amended Type Certificates or Supplemental Type Certificates (STCs) with a certification basis prior to amendment 23-17, where a reciprocating engine is replaced with a turbine engine under § 21.101(b), since it would normally increase the level of safety above that established by the regulations referenced on the original type certificate. An aural speed warning device may be required on such airplanes only if an unsafe condition would exist without one.

b. Procedure. The following procedure should be used to determine whether an unsafe condition exists when converting a reciprocating engine small airplane to turbine engine(s):

(1) Determine if there is any distinctive natural warning that becomes unmistakably evident if the limiting speed (V_{NO}, V_{MO}) is exceeded by 6 knots or the limiting Mach (M_{MO}) is exceeded by .01. Aerodynamic buffet by itself is not considered an adequate natural warning of an overspeed condition (see paragraph 121a(6)).

(2) The airplane should be evaluated for compliance with the high-speed characteristics of § 23.253. An acceptable means of demonstrating compliance is provided in this section.

(3) When evaluating the specific upset conditions for determining the need for an overspeed warning device, record the airspeed at the end of the specified delay times. If an overspeed warning system is not installed, or there is no natural warning, the time delays shall begin at the airspeed defined for overspeed warning, that is, V_{NO}/M_{MO} plus 6 knots or M_{MO} plus 0.01M. If the airplane will accelerate to more than the previous V_{NE} or to a speed more than half way between V_{MO}/M_{MO} and V_D/M_D, then an overspeed warning will be required.

(4) V_{MO}/M_{MO} is established by procedures defined in § 23.335(b) and an upset maneuver that is attitude, time, thrust, and drag sensitive. All established recognition and recovery procedures for determining the acceptability of a given V_{MO}/M_{MO} under §§ 23.251 and 23.253 are predicated upon the pilot receiving a clear and distinct warning slightly past the established V_{MO}/M_{MO}. Definite recognition establishes the beginning of time delay for 3 seconds prior to recovery inititation, thus providing an adequate normal operation margin for recovery. If an airplane accelerates rapidly enough during this time delay, to exceed the previously established V_{NE}, or half the distance between V_{MO}/M_{MO} and V_D/M_D, the time remaining to recover is considered inadequate to meet the criteria envisaged in the procedures prescribed in § 23.335(b). This would be considered as an unsafe feature under the provisions of § 21.21.

289. SECTION 23.1303 (as amended by amendment 23-17) FLIGHT AND NAVIGATION INSTRUMENTS.

a. Free Air Temperature (FAT). Section 23.1303(d) requires that turbine engine-powered airplanes have a free air temperature indicator or an air temperature indicator that provides indications that are convertible to free air. The temperature pickup can be calibrated against a test pickup of known characteristics, or by flying at various speeds at constant altitude, or by tower fly-by. This calibration is frequently done in conjunction with one or more of the airspeed calibration methods described in paragraph 302 of this AC. The constant altitude and tower fly-by calibration methods are described in Air Force Technical Report No. 6273 (see appendix 2, paragraph f(2) of this AC).

b. Remaining Flight and Navigation Instruments. (RESERVED).

290. SECTION 23.1305 (as amended by amendment 23-34) POWERPLANT INSTRUMENTS.

a. Explanation. Section 23.1305 is specific as to the powerplant instruments required for each type of installation. The requirement for specific instruments on specific airplanes should be determined by analysis of type design data prior to certification flight test.

b. Procedures. Verify proper functioning of each required instrument/indicator installed. If the creation of a required malfunction would require establishing a potentially hazardous condition in flight, proper functioning of these indicators may be verified by ground test.

c. Fuel Flowmeters. Advisory Circular (AC) 23.1305-1 covers the installation of fuel flowmeters in airplanes with continuous-flow fuel injection reciprocating engines.

291. SECTION 23.1307 (as amended by amendment 23-23) MISCELLANEOUS EQUIPMENT.

a. Explanation. This subpart requires an approved seat for each occupant, a master switch arrangement, an adequate source of electrical energy and electrical protective devices.

b. Procedures. Verify the proper functioning of each required equipment item. Confirm that when approved production seats are in place, that the seats can be easily adjusted and will remain in a locked position. Confirm that the master switch arrangement is prominently located and marked. Where applicable, proper function of these items may be verified by ground tests.

292. SECTION 23.1309 EQUIPMENT, SYSTEMS, AND INSTALLATIONS. (RESERVED).

293.-299. RESERVED.

Section 2. INSTRUMENTS: INSTALLATION

300. SECTION 23.1321 ARRANGEMENT AND VISIBILITY. (RESERVED).

301. SECTION 23.1322 WARNING, CAUTION, AND ADVISORY LIGHTS. (RESERVED).

302. SECTION 23.1323 (as amended by amendment 23-34) AIRSPEED INDICATING SYSTEM.

a. Explanation.

(1) Airspeed Indicator. An airspeed indicator is usually a pressure gauge that measures the difference between free stream total pressure and static pressure and is usually marked in knots.

(2) Air Data Computer Systems. (RESERVED).

(3) Definitions. Section 1.1 of Part 1 of the FAR defines indicated airspeed (IAS), calibrated airspeed (CAS), equivalent airspeed (EAS), true airspeed (TAS), and Mach number. These definitions include the terms position error, instrument error, and system error, which may need further explanation.

(i) Position Error. Position error is the total-pressure (pitot) and static-pressure errors of the pitot-static installation. By proper design, the total pressure error may be reduced to the point where it is insignificant for most flight conditions. NASA Reference Publication 1046 (see subparagraph g) gives various design considerations. The static pressure error is more difficult to measure and can be quite large.

(ii) Instrument Error. Instrument errors are errors inherent in the instrument for mechanical instruments. These errors are the result of manufacturing tolerances, hysteresis, temperature changes, friction, and inertia of moving parts. For electronic instruments, these errors are due to errors in the electronic element which convert pitot-static pressures into electronic signals. Instrument errors are determined for inflight conditions in steady state conditions. Ground run system calibrations may require the consideration of internal instrument dynamics as would be affected by takeoff acceleration.

(iii) System Error. System error is the combination of position error and instrument error.

(4) Temperatures. Static air temperature (SAT) and total air temperature (TAT) are not defined in section 1.1 of the FAR but may be significant in accurate calibration of airspeed systems. For stabilized values of pressure altitude and calibrated airspeed, TAS is a function of static air temperature. Reference f(2) of appendix 2 discusses the heating effect of the airflow on the temperature sensor and shows how to determine the recovery factor of the sensor. Figure 7 of appendix 7 gives temperature ram rise, if the sensor recovery factor is known.

(5) System Calibration. The airspeed system is calibrated to determine compliance with the requirements of § 23.1323, and to establish an airspeed reference which is used in demonstrating compliance with other applicable regulations. The airspeed system may be calibrated using the speed course method, pacer airplane method, trailing bomb and/or airspeed boom method, tower flyby method, or trailing cone method. The method used will depend on the speed range of the airplane tested, the configuration, and the equipment available. System calibration of the airspeed system is usually determined at altitudes below 10,000 feet. For airplanes approved for flight above 31,000 feet, it is appropriate to verify validity of position errors at the higher operating altitudes. For airplanes where the static ports are located in close proximity to the propeller plane, it should be verified that sudden changes in power do not appreciably change the airspeed calibration. Additionally, for commuter category airplanes, § 23.1323(c) requires an airspeed calibration for use during the accelerate-takeoff ground run.

(6) Instrument Calibration. All instruments used during the test should be calibrated and all calibration curves included in the Type Inspection Report.

b. Speed Course Method. The speed course method consists of using a ground reference to determine variations between indicated airspeed and ground speed of the airplane. See appendix 9 for test procedures and a sample data reduction.

c. Trailing Bomb and/or Airspeed Boom Method. See appendix 9 for procedures, test conditions, and a sample data reduction.

d. Pace Airplane Method. See appendix 9 for test procedures.

e. Tower Flyby. See paragraph 303 for explanation.

f. Ground Run Airspeed System Calibration. The airspeed system is calibrated to show compliance with commuter category requirements of § 23.1323(c) during the accelerate-takeoff ground run, and is used to determine IAS values for various V_1 and V_R speeds. See appendix 9 for definitions, test procedures, and sample data reductions.

g. Other Methods. Other methods of airspeed calibration are described in NASA Reference Publication 1046, "Measurement of Aircraft Speed and Altitude," by W. Gracey, May 1980.

303. SECTION 23.1325 STATIC PRESSURE SYSTEM (as amended by amendment 23-34).

a. Definitions. Paragraph 302 defines several of the terms associated with the pitot-static systems. Others may need further explanation.

(1) Altimeter. An altimeter is a pressure gage that measures the difference between a sea level barometer pressure set on the instrument and static pressure, and indicates in units of feet.

(2) Static Error (error in pressure altitude). The error which results from the difference between the actual ambient pressure and the static pressure measured at the airplane static pressure source is called static error. Static error causes the altimeter to indicate an altitude which is different than actual altitude. It may also affect the errors in the airspeed indicating system.

b. Static System Calibration. The static system is calibrated to determine compliance with the requirements of § 23.1325. The static system may be calibrated by utilizing a trailing bomb, cone, or tower flyby method. Alternately, for properly designed pitot systems, the pitot has minimal effects on the airspeed position error (dV_c), as determined for § 23.1323. For these systems, static error (dh) may be calculated by the following equation:

$$dh = .08865\left(dV_c\right)\left\{1 + .2\left\{\frac{V_c}{661.5}\right\}^2\right\}^{2.5}\left\{\frac{V_c}{\sigma}\right\}, \text{ ft.}$$

where V_c = calibrated airspeed, knots

σ = ambient air density ratio

dV_c = airspeed position error

c. Test Methods. The methods specified for calibration of the airspeed indicating systems, including test conditions and procedures apply equally for determining static error and error in indicated pressure altitude, and are usually determined from the same tests and data.

d. Tower Flyby. The tower flyby method is one of the methods which results in a direct determination of static error in indicated pressure altitude without the need for calculating from airspeed position error.

e. Procedures and Test Conditions for Tower Flyby.

(1) Air Quality. Smooth, stable air is needed for determining the error in pressure altitude.

(2) Weight and C.G. Same as for calibrations of the airspeed indicating system.

(3) Speed Range. The calibration should range from 1.3 V_{SO} to 1.8 V_{S1}. Higher speeds up to V_{MO} or V_{NE} are usually investigated so that errors can be included in the AFM for a full range of airspeeds.

(4) Test Procedures.

(i) Stabilize the airplane in level flight at a height which is level with the cab of a tower, or along a runway while maintaining a constant height of 50 to 100 feet by use of a radio altimeter. A ground observer should be stationed in the tower, or on the runway with an altimeter of known instrument error. Pressure altitude is recorded on the ground and in the airplane at the instant the airplane passes the ground observer.

(ii) Repeat step (i) at various airspeeds in increments sufficient to cover the required range and at each required flap setting.

(5) Data Acquisition. Data to be recorded at each test point:

(i) Airplane IAS.

(ii) Airplane indicated pressure altitude.

(iii) Ground observer indicated pressure altitude.

(iv) Radar altimeter indication (if flown along a runway).

(v) Wing flap position.

(vi) Landing gear position.

(6) Data Reduction.

(i) Method.

(A) Correct indicated pressure altitude values for instrument error associated with each instrument.

(B) To obtain test pressure altitude, adjust the ground observed pressure altitude by the height read from the radar altimeter. No adjustment is required if the airplane was essentially the same level as the ground operator (tower cab). Static errors may be adjusted from test pressure altitude to sea level by the following:

$$dh_{(S.L.)} = \left\{ dh_{(TEST)} \right\} \left\{ \sigma(TEST) \right\}$$

Where:

$dh_{(TEST)}$ = Difference in test pressure altitude and airplane pressure altitude with associated instrument errors removed.

σ (TEST) = ambient air density ratio.

(ii) Plotting. Static error at sea level ($dh_{(S.L.)}$) should be plotted vs. test calibrated airspeeds.

(7) Required Accuracy. Section 23.1325(e) requires that the error in pressure altitude at sea level (with instrument error removed) must fall within a band of $\pm$ 30 feet at 100 knots or less, with linear variation of $\pm$ 30 feet per 100 knots at higher speeds. These limits apply for all flap settings and airspeeds from 1.3 V_{SO} up to 1.8 V_{S1}. For commuter category airplanes, § 23.1325(f) requires that the altimeter system calibration be shown in the AFM.

304. SECTION 23.1327 MAGNETIC DIRECTION INDICATOR. (RESERVED).

305. SECTION 23.1329 (as amended by amendment 23-23) AUTOMATIC PILOT SYSTEM. This subject will be covered in a separate AC.

306. SECTION 23.1331 INSTRUMENTS USING A POWER SUPPLY. (RESERVED).

307. SECTION 23.1335 FLIGHT DIRECTOR SYSTEMS. (RESERVED).

308. SECTION 23.1337 (as amended by amendment 23-18) POWERPLANT INSTRUMENTS.

a. Explanation.

(1) Fuel Quantity Indicator. The indicator should be legible and easily read without excessive head movement. The calibration units (pounds or gallons) and the scale graduations should be readily apparent. Units should be consistent with AFM procedures and performance data.

(2) Auxiliary Tanks. A fuel quantity indicator is not required for a small auxiliary tank that is used only to transfer fuel to another tank if the relative size of the tank, the rate of fuel flow, and operating instructions are adequate. The requirement for a separate quantity indicator should be determined by analysis of design data prior to flight test. The relative size of the tanks, intended use of the auxiliary tanks, complexity of the fuel system, etc., should be considered in determining the need for a fuel quantity indicator. If an indicator is not installed, flight manual procedures should ensure that once transfer of fuel is started, all fuel from the selected auxiliary tank can be transferred to the main tank without overflow or overpressure.

b. Procedures. Evaluate indicators for clarity and legibility. Review AFM for consistency of units and validity of procedures.

309.-318. RESERVED.

Section 3. ELECTRICAL SYSTEMS AND EQUIPMENT

319. SECTION 23.1351 GENERAL. (RESERVED).

320. SECTION 23.1353 STORAGE BATTERY DESIGN AND INSTALLATION. (RESERVED).

321. SECTION 23.1357 CIRCUIT PROTECTIVE DEVICES. (RESERVED).

322. SECTION 23.1361 MASTER SWITCH ARRANGEMENT. (RESERVED).

323. SECTION 23.1367 SWITCHES. (RESERVED).

324.-328. RESERVED.

Section 4. LIGHTS

329. SECTION 23.1381 INSTRUMENT LIGHTS. (RESERVED).

330. SECTION 23.1383 LANDING LIGHTS. (RESERVED).

331.-335. RESERVED.

Section 5. SAFETY EQUIPMENT

336. SECTION 23.1411 GENERAL. (RESERVED).

337. SECTION 23.1415 DITCHING EQUIPMENT. (RESERVED).

338. SECTION 23.1416 (as added by amendment 23-23) PNEUMATIC DEICER BOOT SYSTEM. See AC 23.1419-1.

339. SECTION 23.1419 (as amended by amendment 23-14) ICE PROTECTION. See AC 23.1419-1.

340.-349. RESERVED.

Section 6. MISCELLANEOUS EQUIPMENT

350. SECTION 23.1431 ELECTRONIC EQUIPMENT. (RESERVED).

351. SECTION 23.1435 HYDRAULIC SYSTEMS. (RESERVED).

352. SECTION 23.1441 OXYGEN EQUIPMENT AND SUPPLY. (RESERVED).

353. SECTION 23.1447 EQUIPMENT STANDARDS FOR OXYGEN DISPENSING UNITS. (RESERVED).

354. SECTION 23.1449 MEANS FOR DETERMINING USE OF OXYGEN. (RESERVED).

355.-364. RESERVED.